PSD 69 03
Copy 1

CR73313
Available to the Public

The Pennsylvania State University
The Graduate School
King of Prussia Graduate Center

# Biosatellite Attitude Stabilization and Control System

A report in
Engineering Science
by

Francis A. Calabrese

Submitted in partial fulfillment
of the requirements
for the degree of

Master of Engineering

June 1969



Approved:

Report Advisor

Assistant Dean, The Graduate School

# BIOSATELLITE ATTITUDE STABILIZATION AND CONTROL SYSTEM *

This paper will present the Attitude Stabilization and Control System for NASA's earth orbiting Biosatellite. The object of the Biosatellite experiment is to show extra-terrestial effects upon the life cycle of certain plant and animal life.

The Attitude Stabilization and Control System is required to control the effects of gravity, and to align the vehicle attitude prior to retro fire. The operational description of this system will be included, showing the control technique employed and the block diagram development for each mode of operation. The system description will include all sensors employed in gathering the required body information for accurate vehicle control, and the system transfer function diagram and stability analyses. The vehicle modes of operation will be defined with an explanation for each. This paper will also cover the analytical aspects of the design, and will specify the guiding equations utilized. Special case analyses will be included, showing the effects of natural phenomena upon vehicle response.

Since the fundamental operation of the control system is non-linear, single axis phase plane analyses will be performed to describe the stability of the system. Critical factors of the analysis will be identified, such as acquisition time and gas consumption for the two basic modes of operation. The optimization of the torque level and switching lines will be discussed in light of the vehicle requirements.

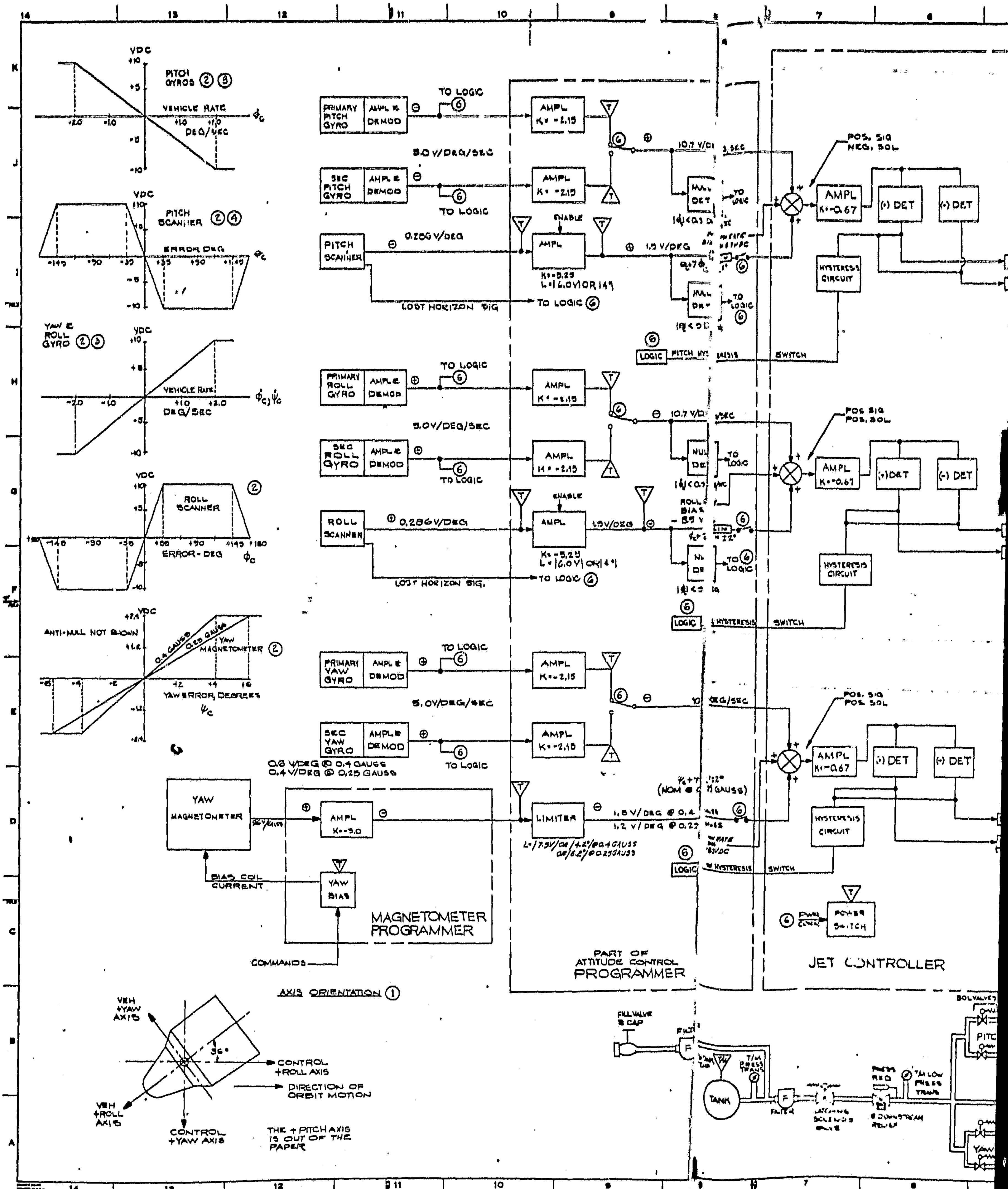
In addition, the analytical studies will be compared with the results of the hybrid simulation, which gives consideration to the second order effects upon vehicle performance.

* Work is being performed on Contract NAS2-1900 from the National Aeronautics and Space Administration, Ames Research Center, Moffett Field, Calif.

# Table of Contents

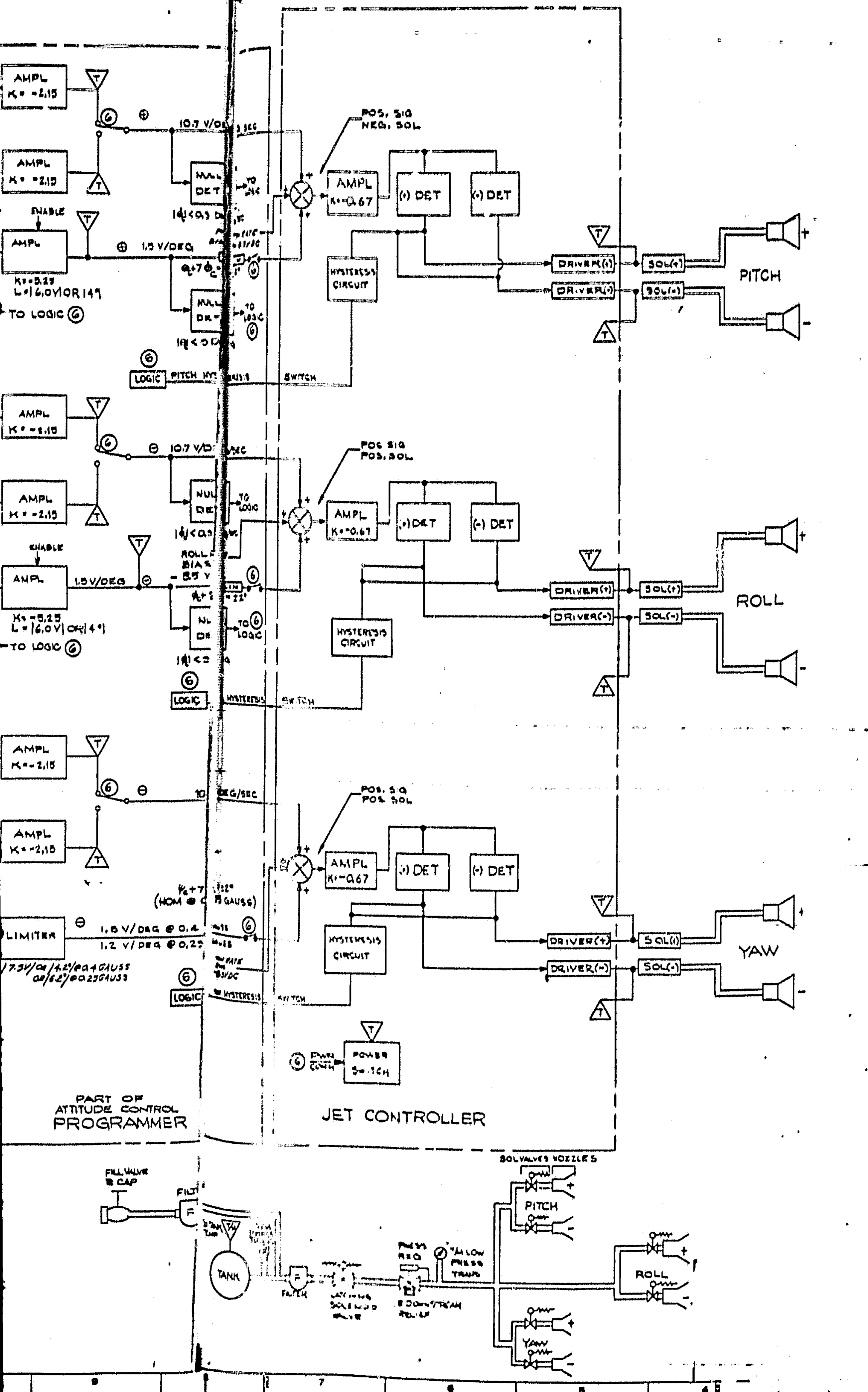
FIGURE 2-A

FIGURE




BIOSATELLITE ATTITUDE CONT
BLOCK DIAGRAM

FIGURE - 1-B




REVISIONS
A PITCH AND ...

LEGEND

(T) = TELEMETRY PROCESSING CIRCUITS.

⊕ = POLARITY OF SIGNAL IS THE SAME AS SIGN OF ERROR.

⊖ POLARITY OF SIGNAL IS OPPOSITE TO SIGN OF ERROR.

NOTES:

1. AXIS REFERENCE SYSTEMS ARE BOTH RIGHT HAND.
2. ALL SENSORS ARE ALIGNED TO CONTROL AXIS.
3. GYRO SIGNAL POLARITY IS OBTAINED BY PROPER ORIENTATION OF THE GYROS IN THE VEHICLE.
4. PITCH IR SENSOR SCAN AXIS IS THE SAME SENSE AS THE MINUS PITCH AXIS.
5. NOZZLES ARE LOCATED WITH RESPECT TO CONTROL AXES.
6. ALSO SEE LOGIC DIAGRAM 47R196084.
7. HIGH HYSTERESIS IN RATE CONTROL MODE NOZZLE SOLENOIDS OPEN AT 3.0V (0.28 DEG/SEC) AND CLOSE AT 1.0 V (0.09 DEG/SEC). LOW HYSTERESIS IN DE-ORBIT MODE NOZZLE SOLENOIDS OPEN AT 3.0V AND CLOSE AT 2.9V. (ALL VALUES ARE NOMINAL).
8. THE PRIMARY AND SECONDARY RATE AND POSITION CHANNELS HAVE TWO DETECTORS (ONE POSITIVE DETECTOR AND ONE NEGATIVE DETECTOR) IN BOTH ROLL AND PITCH. THERE ARE NO DETECTORS IN THE YAW CHANNELS.

47E187578

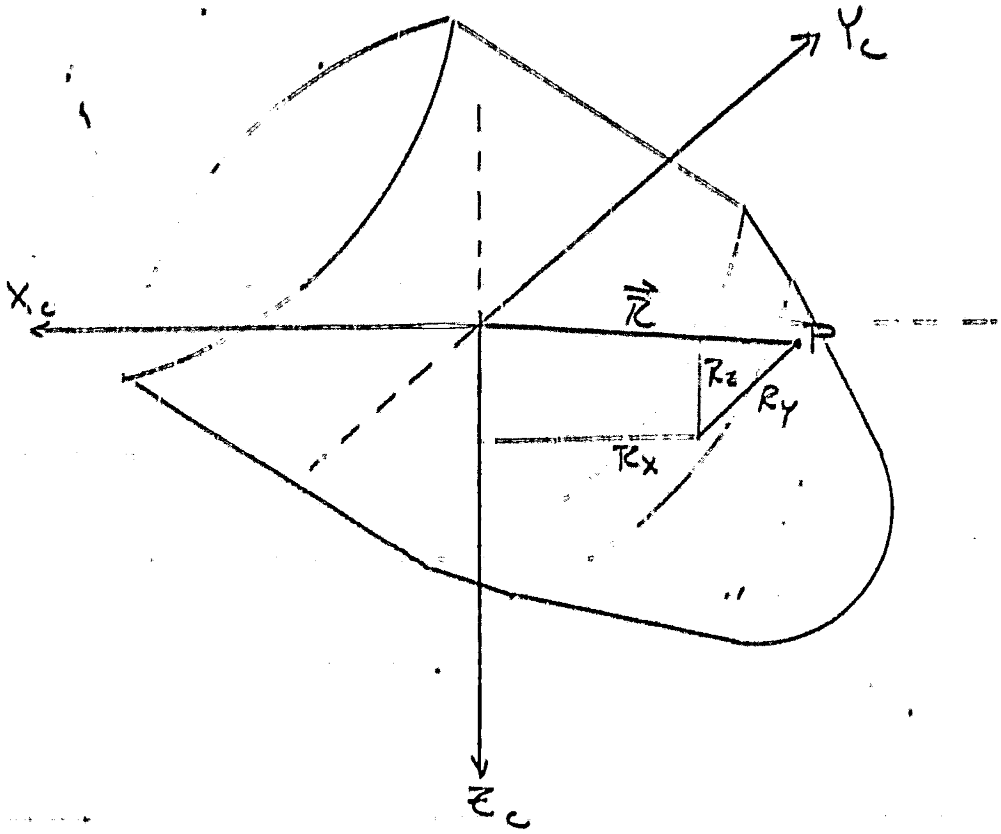
BIOSATELLITE ATTITUDE CONTROL SYSTEM BLOCK DIAGRAM

FIGURE 1

# REQUIREMENTS

Biosatellite is an earth orbiting biological satellite designed to operate for extended periods of time outside the limits of the earth's gravitational field. The purpose of Biosatellite "[redacted]" is to determine the effects of radiation and weightlessness upon the life cycle of plant and animal life. In keeping with the mission objectives, the attitude control system (ACS) on board is required to maintain the acceleration forces on the experiments within $10^{-5}$ G for 95% of the orbital flight, and not to exceed the absolute limit of $0.95 \times 10^{-4}$ G. In addition, at the end of the flight, the ACS is required to aligh the vehicle in the proper attitude for retrofire and eventual vehicle recovery.

The attitude control system on board the spacecraft is a closed loop rate damped nonlinear control system which is required to stabilize the spacecraft in both a rate mode and a position mode during the respective two phases of the exospheric flight. The block diagram of the ACS is shown below in figure 1 . The maximum allowable rate limit was established to comply with the acceleration requirements of the mission, and this criteria was determined by studying the dynamics of rigid bodies.




1

1. $\vec{a} = \ddot{\vec{R}} + \vec{\omega} \times (\vec{\omega} \times \vec{R}) + \dot{\vec{\omega}} \times \vec{R} + 2\vec{\omega} \times \dot{\vec{R}}$

SINCE P IS A FIXED POINT

$$\ddot{\vec{R}} = \dot{\vec{R}} = 0$$

2. $\vec{a} = \vec{\omega} \times (\vec{\omega} \times \vec{R}) + \dot{\vec{\omega}} \times \vec{R}$

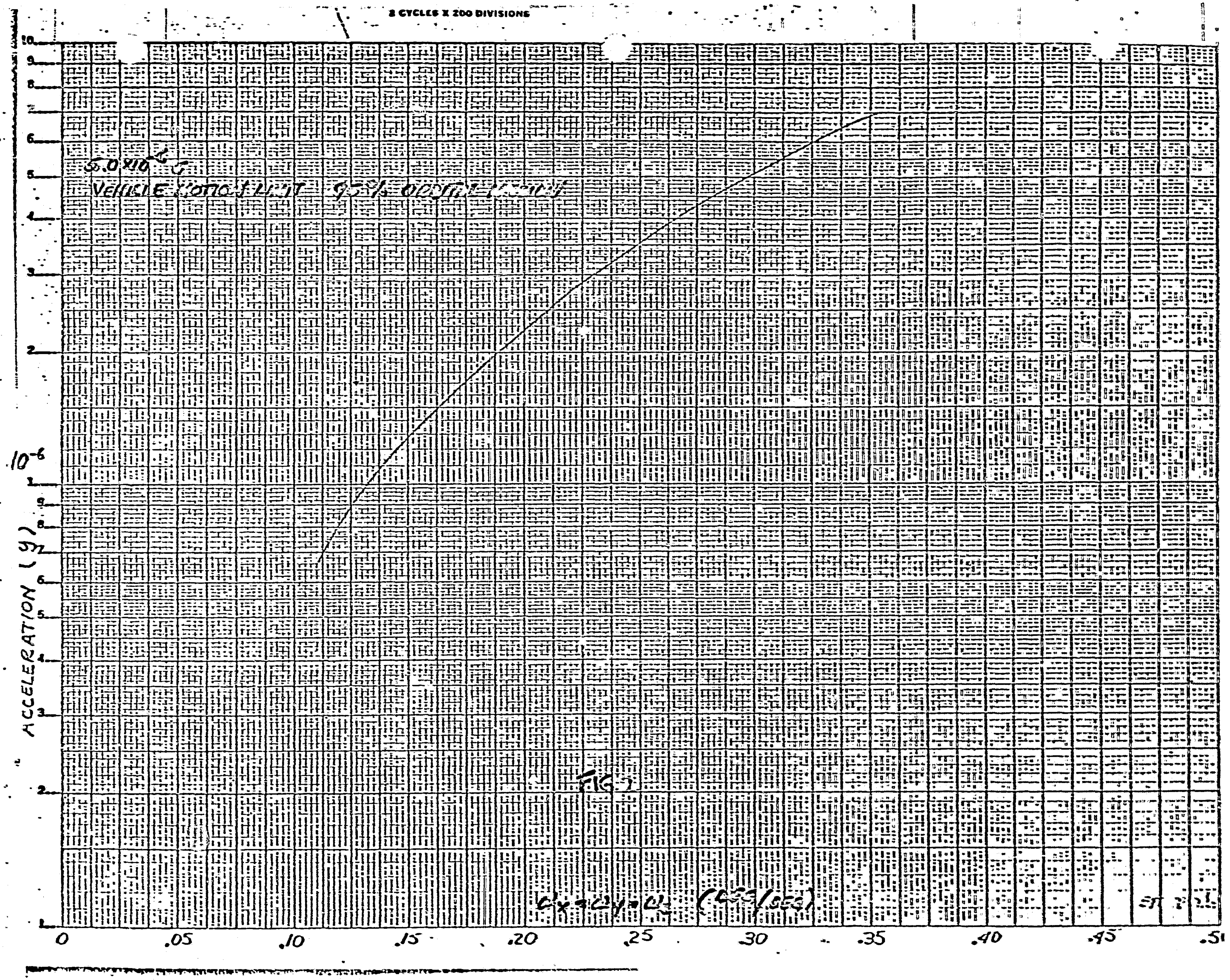
RESOLVING THE VECTOR INTO COMPONENTS

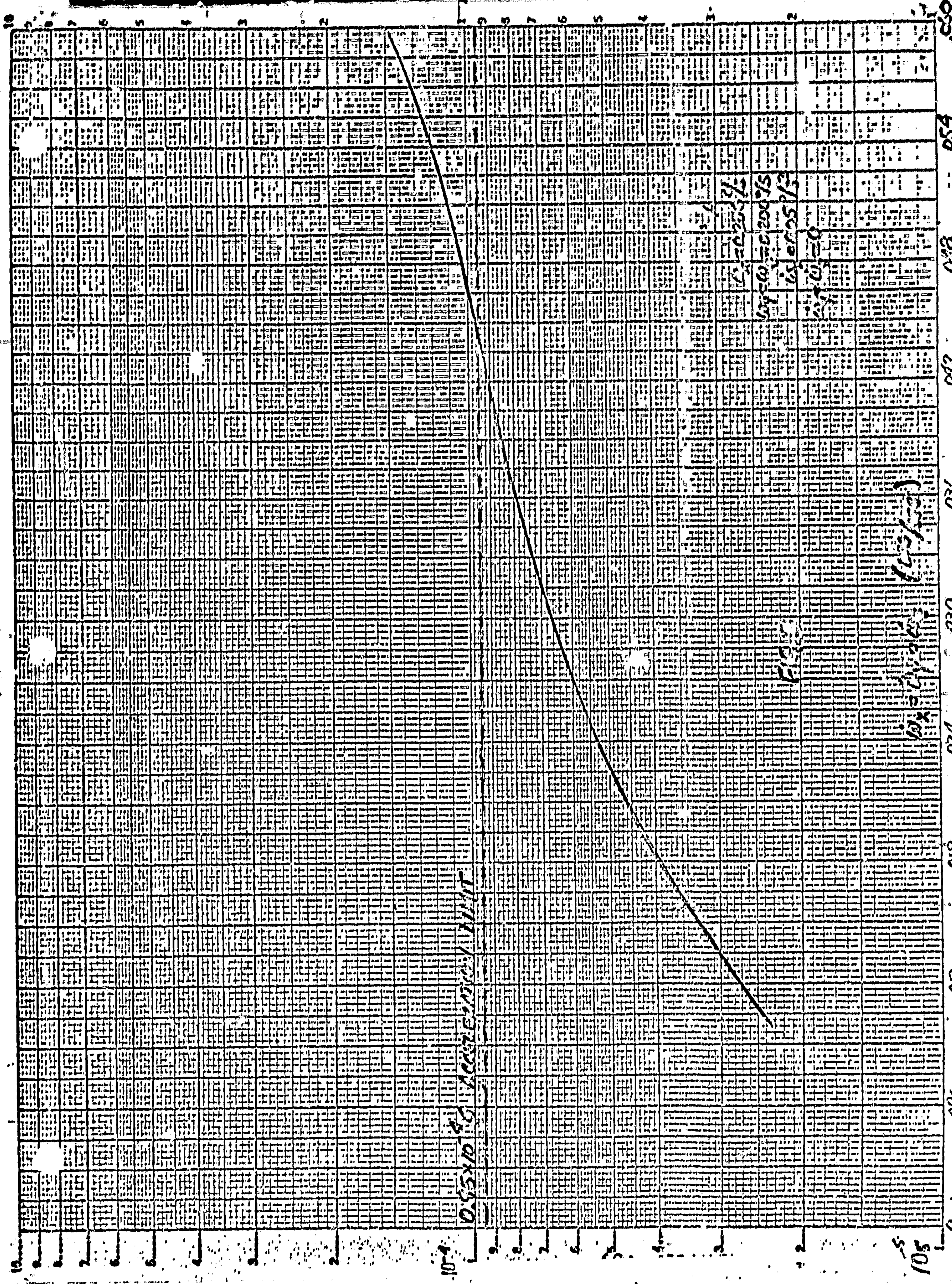
$$a_x = -R_x(\omega_y^2 + \omega_z^2) + R_y(\omega_x\omega_y - \dot{\omega}_z) + R_z(\omega_x\omega_z + \dot{\omega}_y)$$

$$a_y = -R_y(\omega_z^2 + \omega_x^2) + R_z(\omega_y\omega_z - \dot{\omega}_x) + R_x(\omega_x\omega_y + \dot{\omega}_z)$$

$$a_z = -R_z(\omega_x^2 + \omega_y^2) + R_x(\omega_x\omega_z - \dot{\omega}_y) + R_y(\omega_y\omega_z + \dot{\omega}_x)$$

3. $a = \sqrt{a_x^2 + a_y^2 + a_z^2}$


2 CYCLES X 200 DIVISIONS
ACCELERATION (g)
$10^{-6}$
0
.05
.10
.15
.20
.25
.30
.35
.40
.45
.51



0.05x10^-4 G ACCELERATION LIMIT
10^-4
10^-5
0
.006
.012
.018
.024
.030
.036
.042
.048
.054
.060

The above equations were evaluated at the experiment location (Rx=-1.9?; Ry=0.707, Rz=0.888) for values of $\omega$ and $\dot{\omega}$ varying between 0.115-0.40 degrees/sec and 0.0-0.06 degrees/sec$^2$, respeitvely, to ascertain combinations of each which would satisfy the G requirements. It was decided that the rotational disturbance accelerations upon the uncontrolled vehicle would not be allowed to exceed 0.5 X $10^{-5}$ G. By applying this criteria, figure 2 (plot of equation 3 for spin rate) establishes the maximum allowable rate about each of the three axes at approximately 0.286 degrees/sec. When the vehicle rates are being controlled about all three axes (a worst case condition), figure 3 establishes the control acceleration at 0.045 degrees/sec$^2$. The rate limit will now be utilized as the control threshold, which is restrained by pneumatically induced control torques acting about each of the vehicle axes.

RATE MODE

While operating in the rate mode, the vehicle position is allowed to drift and only the vehicle rate error is nulled to within the ACS rate threshold of o.286 degrees/sec. Shown in figure 4 is the block diagram of a single channel of the system while operating in this mode. The rate error signal of the vehicle is sensed by the rate gyro. This signal is amplified in the attitude control programmer, which receives all signals from the sensors and telemetry to be utilized in the ACS. The amplified signal then goes to the jet controller where it is fed into a threshold detector. If the error

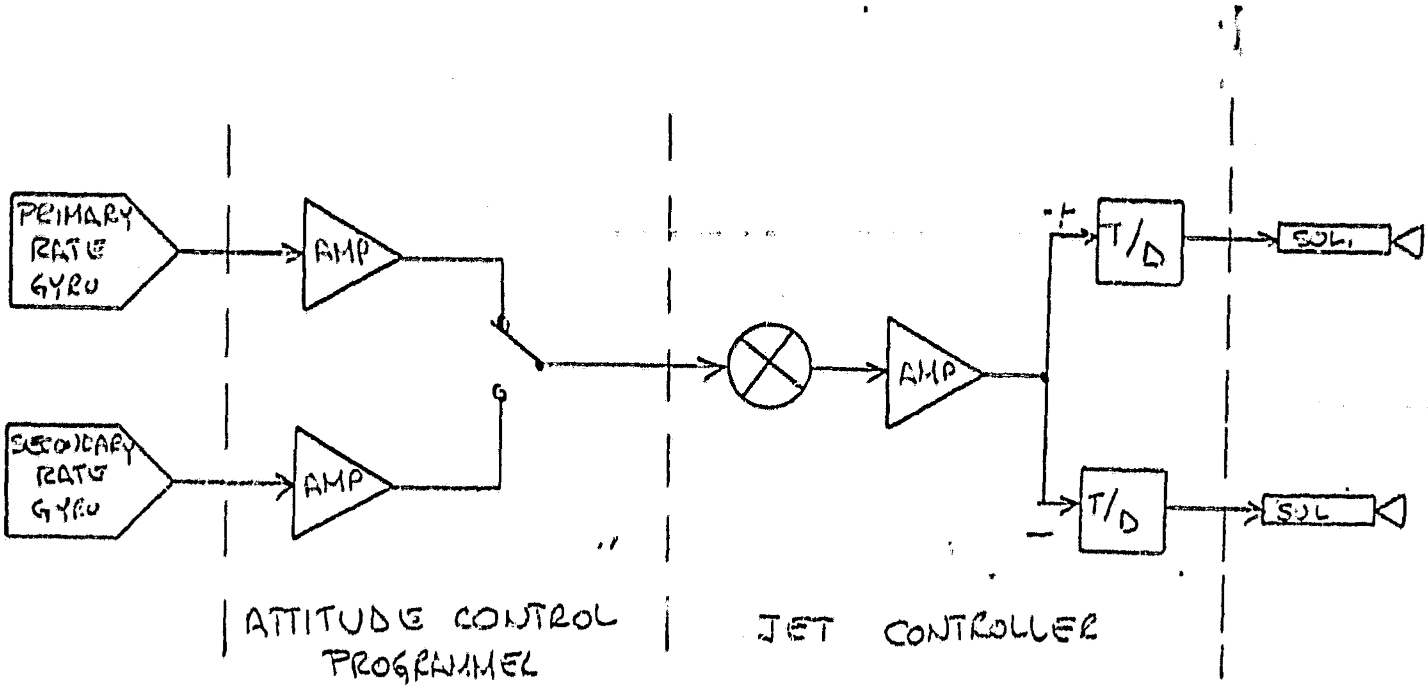



FIGURE 4 - RATE CONTROL BLOCK DIAGRAM

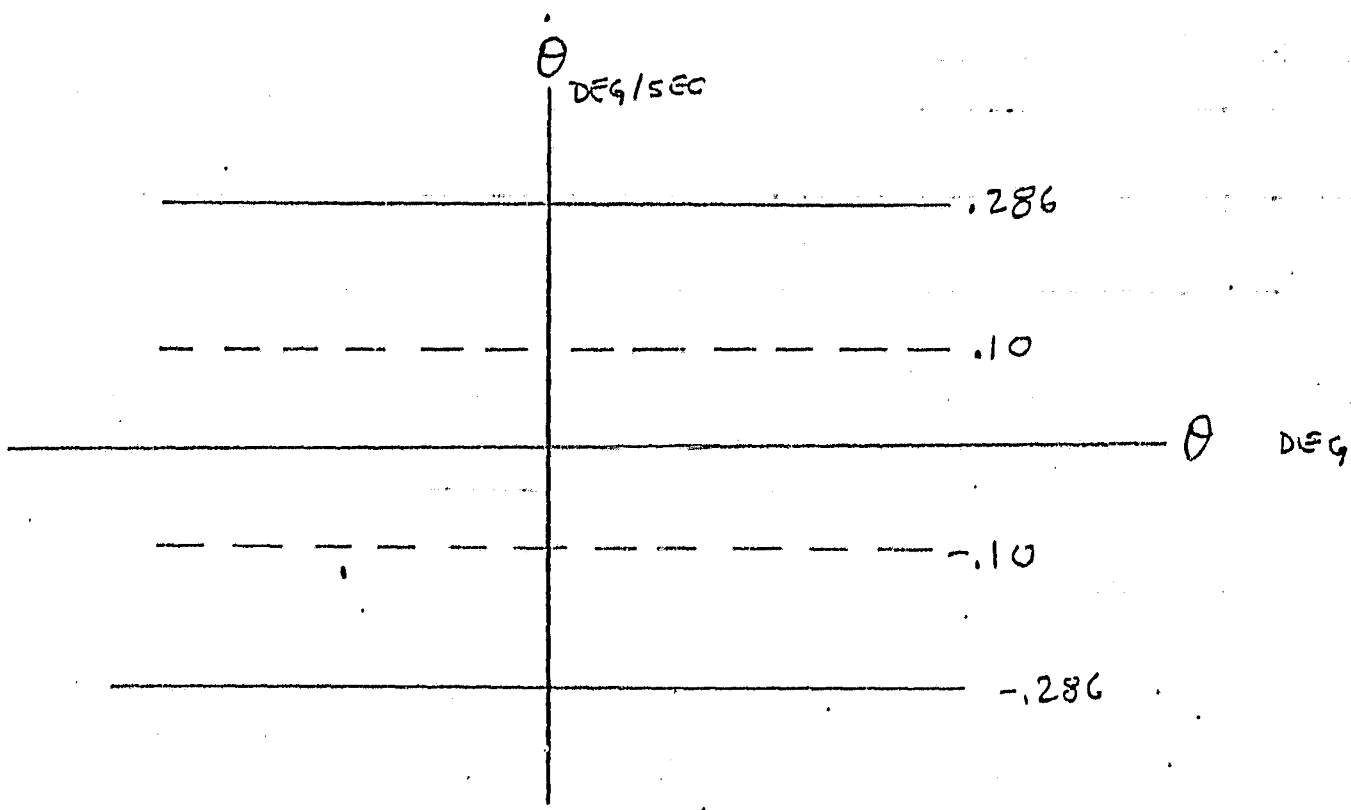



FIGURE 5 - RATE CONTROL SWITCHING LINES

signal is of a sufficient amplitude, the threshold detector operates the appropiate solenoid valve to reduce the vehicle rate by releasing the prescribed amount of $N_2/CF_4$ gaseous mixture. This type of control is either full on or off, thus receiving the mane "Bang-Bang" nonlinear control system which may be analyzed on a phase plane (see appendix A), and shown to produce stable rate control for the vehicle. To prevent the system from "chattering," repeatedly cycling on and off whenever the rates exceed the threshold and are summarily reduced below the threshold, hysteresis is built into the control loop. This eliminates pneumatic sputter* and improves the efficiency and performance of the limit cycle. The switching lines utilized in the rate control mode are shown below in figure 5.

However, there is one drawback with systems of this type. The amplifiers, threshold detectors, solenoid drivers, sensors, logic and other circuits consume large quantities of electrical power. For missions of extended duration electrical power must be conserved, and thus necessitated the need for automatic rate control mode. While operating in this mode, and when the rates are below the threshold, all the ACS electronics are off, except for some sensing circuits. However, once the rate threshold is reached the control system electronics (ASSUMING NEGLIGABLE TIME DELAYS) are triggered on by the rate gyros, and control is resumed in the usual fashion. The hystersis built into the control system, as shown in the

* SPUTTER IS THE UNDESIRABLE SPRAYING OF GAS IN AN UNCONTROLLED FASHION. IT IS INDUCED BY RAPID CYCLING OF THE SOLENOID VALVE, WHERE THE RESPONSE TIME OF THE SOLENOID VALVE EXCEEDS THE DELAY TIME OF THE PNEUMATIC EXHAUST.

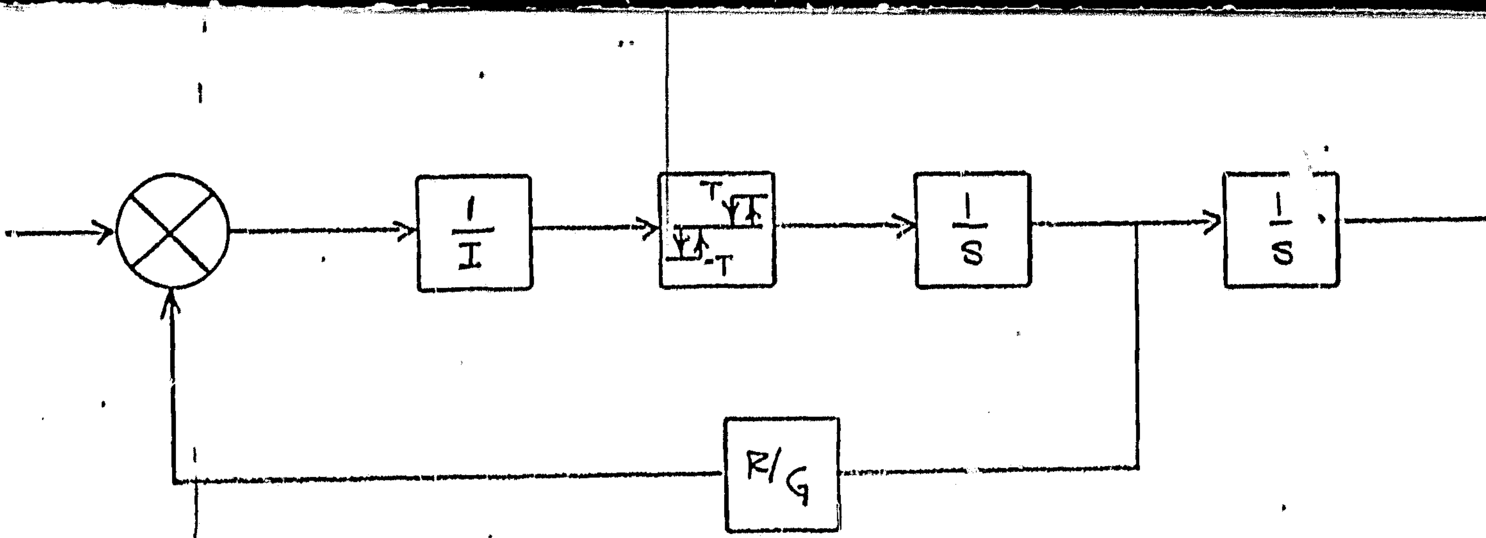



FIGURE 6 - CONTROL DIAGRAM, RATE MODE

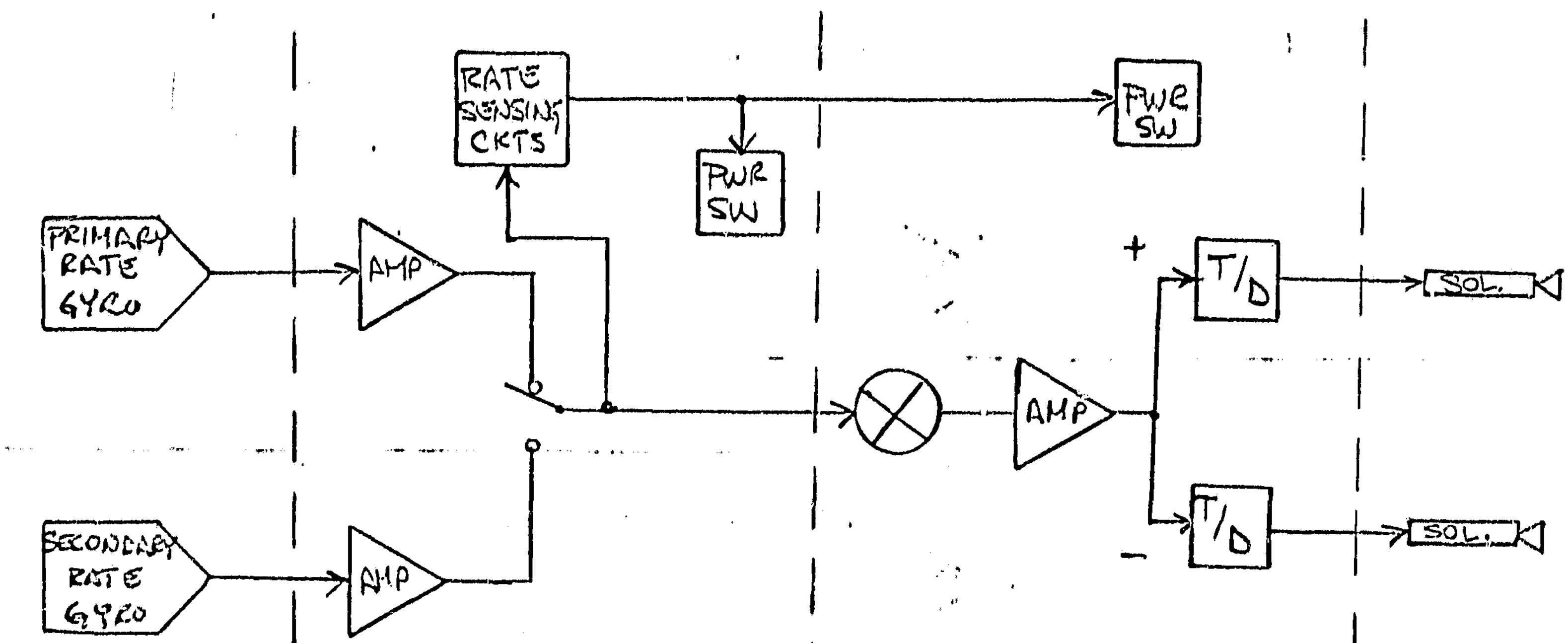



FIGURE 7 - RATE CONTROL BLOCK DIAGRAM WITH AUTOMATIC RATE CONTROL.

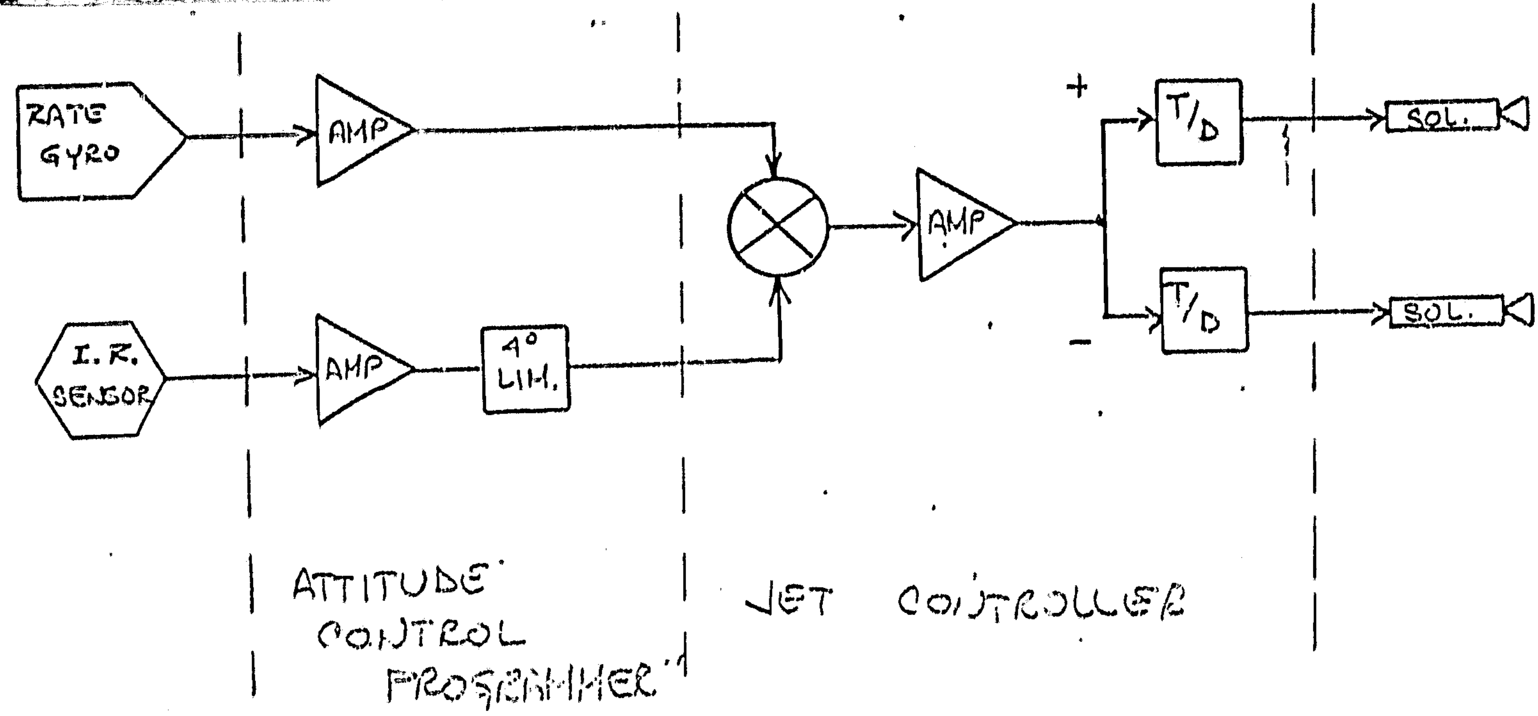



FIGURE 8 - POSITION MODE BLOCK DIAGRAM

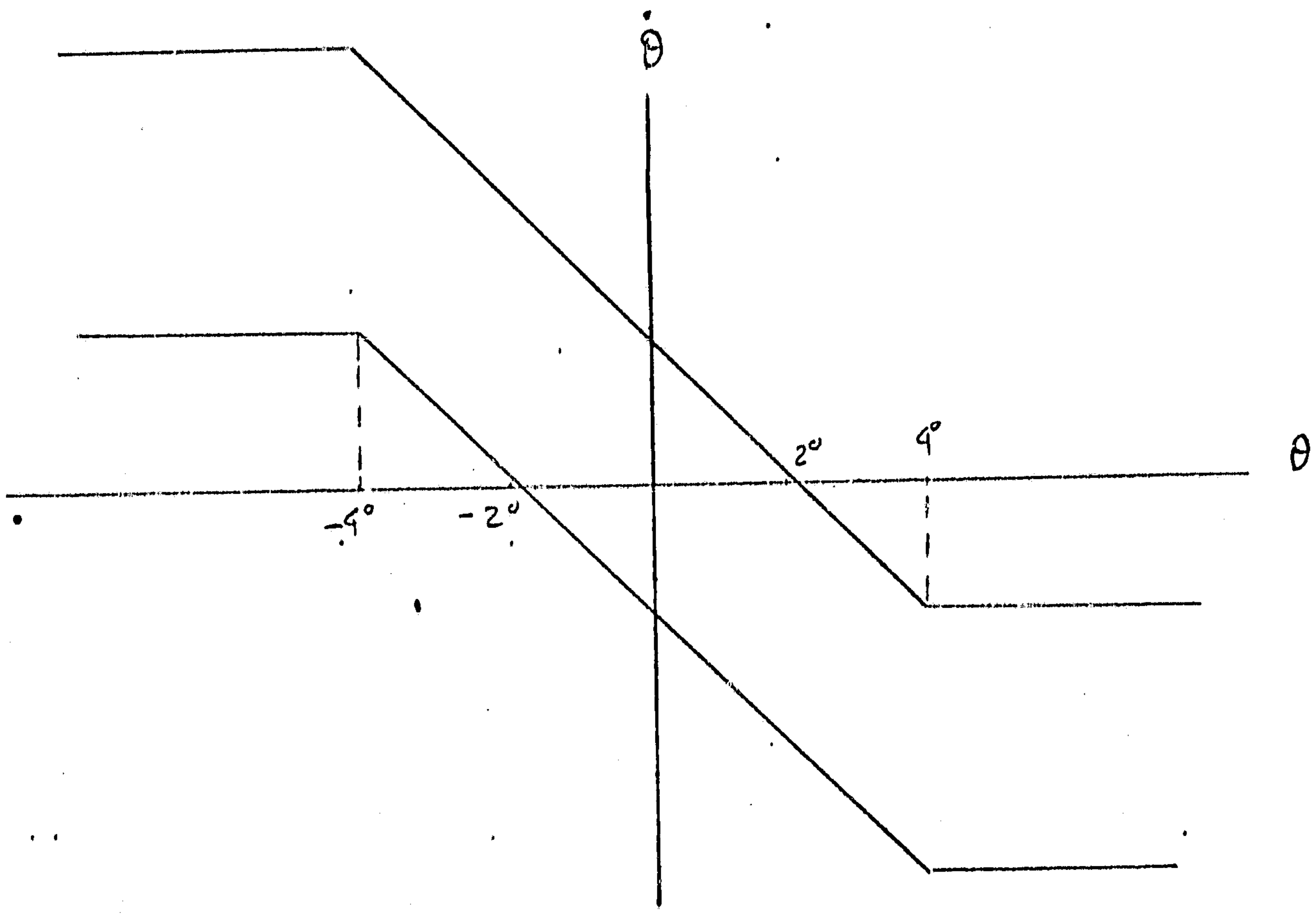



FIGURE 9 - POSITION MODE SWITCHING LINES

control diagram, figure 6, also presvents the ACS from switching back and forth between high and low power. The block diagram of the rate control mode is modified as shown in figure 7.

POSITION MODE

The position mode of operation is required to stabilize the spacecraft in the proper attitude for retrofire. The "Deorbit Mode" as it is referred to, is commanded via telemetry at the end of the mission. The vehicle is again controlled by the same bang-bang thrusters as in the rate control mode. HOwever, the control equation is modified to correct for position errors. The position error signal obtained from the attitude sensor is summed with the rate error signal. Both of which are processed in the ACP before they are summed in the jet controller. AS shown in figure 8, after the signals are summed, they are also amplified in the jet controller, and corried to the T/D. TRANSMITTED If the signal is of sufficient magnitude the T/D operates the proper solenoid valve until the position error is nulled. The control equation which directs the vehicle motion in the position mode is shown below.

$$\theta + 7\dot{\theta} = 2^{\circ}$$

The deadband of $2^{\circ}$ is selected based upon the alottment of tolerable error to each component fo the subsystem (established by the subsystem error analysis), and the required pointing accuracy for successful deorbit and recovery. The gain of 7 sec. in the rate loop over the position loop was selected to minimize the acquisition time and maintain the rate threshold. The switching lines for this equation are shown in figure 9. The slope of the unsaturated switching line is -1/7. At $4^\circ$ the position signal is saturated, this prevents the vehicle rates from becoming excessive, possibly reducing the system performance by causing the gas consumption to increase during acquisition, and also maintains the rate limit at 0.286 deg/sec. The rate limits, determined from the position saturation of the control equation, are shown below.

$$\theta_{sat} + 7\dot{\theta} = \pm 2^\circ \qquad \dot{\theta} = \text{vehicle rate}$$

$$\dot{\theta} = \frac{\pm 2 - \theta_{sat}}{7} \qquad \theta = \text{vehicle position}$$

$$\theta_{SAT} = \text{position saturation}$$

@

$\theta_{sat} = \pm 4^\circ$ DB = $2^\circ$ $\dot{\theta}$ = -2/7, 6/7 respectively

$\theta'_{sat} = \pm 4^\circ$ DB = $-2^\circ$ $\dot{\theta}$ = -6/7, 2/7 respectively

Once acquisition is obtained the vehicle will limit cycle about $\pm 2^\circ$ (see appendix A).

Each of the three axes is controlled independently of the other in both the rate control mode and the Deorbit ~~attitude~~ Mode. However, during the the deorbit attitude acquisition both roll and pitch must be acquired before yaw control is initiated. This requirement is necessary because of the nature of the position sensors utilized in the control system. The roll and the pitch position sensors are infared horizon sensors, which null their respective vehicle axes errors with respect to the earth. This establishes a plane ORTHOGONAL TO A LOCAL VERTICAL in which the vehicle may rotate. In order to obtain the third axis, DESIRED ORIENTATION IN yaw, a magnetometer is employed to sense the earth's magnetic field. Since the magnetic field intensity and declination (dip) angle varies widely over the earth, only the horizontal component is utilized for vehicle control. This necessitates the magnetometer being in the horizontal plane before yaw control may be obtained.

The arrangement of this type of sensor system provides the vehicle with the capability of deorbiting anywhere along its trajectory PROVIDED THE VALUE OF THE MAGNETIC FIELD IS DEFINED. Of course, certain areas are more desirable than others, PARTICULARLY, SINCE THE REENTRY VEHICLE IS TO BE RECOVERED BY AIR SNATCH OVER THE PACIFIC. THEREFORE, IT IS ADVANTAGEOUS TO SELECT A DEORBIT POINT, SUCH THAT THE RECOVERY ZONE IS WITHIN THE RANGE OF THE RECOVERY AIRCRAFT, AND BACK UP SURFACE SHIP.

ATTITUDE CONTROL SENSORS

Rate Sensor- The Biosatellite three axis rate sensor package is a triad of single axis rate sensing gyros mounted such that their input axes are orthogonal to each other, and in the same coordinate system as the vehicle control axes. Each gyro is contrained to one degree of freedom with its output linearly proportional to the rate input.

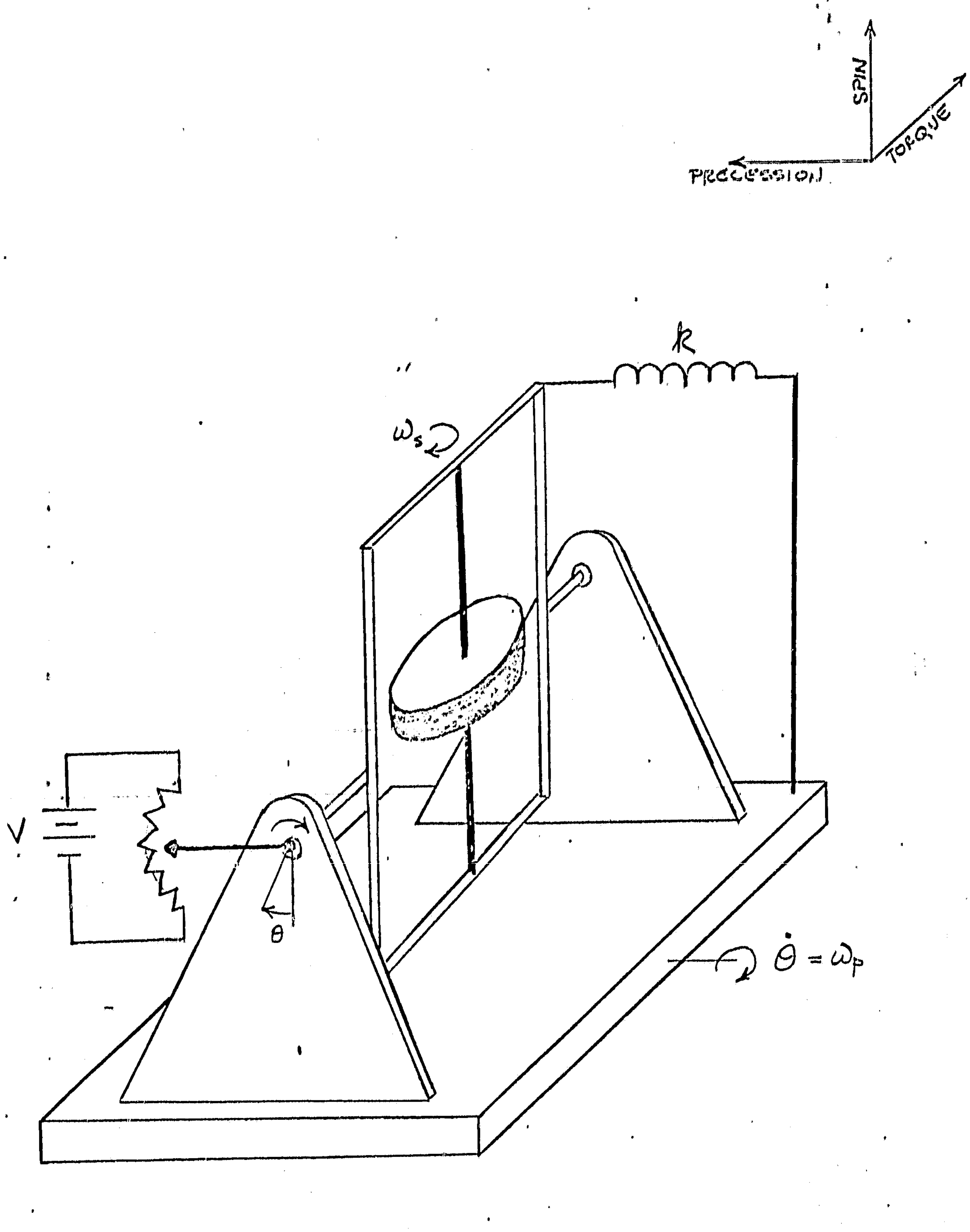



FIGURE 10 - RATE GYRO

2

A rate gyro is essentially a rotating wheel mounted in a frame called the gimbal. As illustrated in figure 10, the gimbal of a single degree of freedom gyro rotates about only one axis with respect to the frame. The fundamental principle of a gyroscope is that ITS SPINNING MASS prefers to stay fixed within its coordinate system. If an external torque were applied to the gimbal, a precession would result about the base. This phenomena is defined by the relationship-

$$T = I \times \omega_S \times \omega_P$$

I - inertia of the gyro wheel

$\omega_S$ - spin rate applied to wheel

$\omega_P$- precession rate

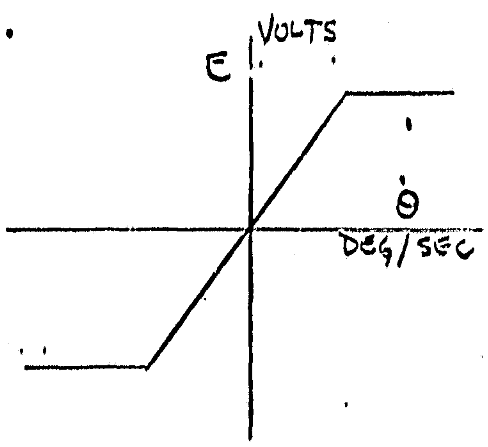
In the case of a rate gyro, the precession axis is the input axis, and the torque measured is therefore proportional to this precession rate. A convient way of measuring this torque is to restrain the gimbal with a spring, and the deflection of the spring will be proportional to the torque. This deflection, in turn, is then measured electrically with a potentiometer or similiar pickoff.

To summarize the operations of a gyro:

1. V ehicle rates are input about the gyro precession axis.
2. The precession is then converted to a torque by the gyro.
3. The torque is proportional to the linear deflection of the gimbal restraining spring.

4. The spring deflection is measured with a pickoff.

Hence the spring restrained rate gyro sensor, mounted such that the precession axis is parallel to the axis about which the rate is to be measured, will convert this input rate to an electrical output. ~~which is~~ THIS SIGNAL IS utilized to rate stabilize and control space vehicles. An instrument of this type, of course, has certain restrictions to assure linear operation. The bearings must have an extremely low frictional level so as not to impede the motion of either the gimbal or the wheel. The wheel is critically balanced so that the cross products of inertia are ~~approximately~~ EFFECTIVELY zero. The deflection of the gimbal is restricted by mechanical stops, which ~~confines~~ CONTRAINS the instrument to both operate in the linear range of the spring, and to maintain a small angular deflection such that $\sin \theta = \theta$. The input-output relationship of the gyro is shown below. The scale factor (volts/deg/sec) is determined by adjusting the spring




constant, k, of the gimbal restraining spring. This output signal is in turn amplified, filtered and transmitted to the attitude control programmer.

Infared Horizon Sensor - The IR sensors will locate the horizon of th earth by detecting the differential temperature between cold space and earth. Each sensor contains a rotating prism with a $2^{o} \times 8^{o}$ field of view. The rotation of the prism scribes a conical surface with a $110^{o}$ apex angle. The sensor is oriented such that its field of view scans both the earth and space as the prism sweeps through a complete rotation.

Before the IR sensor may gather the necessary position information and be utilized in the control loop, it is necessary to establish earth presence. This is the threshold which defines the percentage of the earth to sky being scanned. Too small or too large a pulse will produce poor signal resolution. A low signal earth pulse where earth-sky contrast is greatly reduced, makes noise a limiting factor. Each of these factors, in addition, to excessive sunlight determine the criteria for earth presence.

To obtain attitude information, INFRARED energy IN THE 8.5-20μ SPECTRAL REGION, which is naturally radiated from the earth and its atmosphere, is optically collected during the scan. This energy is then focused on an infared sensitive detector which produces a voltage that is related to the temperature of the area scanned. This detector is a thermistor bolometer. The bolometer is a transducer which converts incident infared energy into an electrical signal which is proportional to the radiant intensity. The output voltage will then appear as a square wave, as the prism alternately detects the earth and space during the scan. The pulse width of the square wave is proportional to the time of the earth scan. Vehicle attitude deviation with respect to the earth is derived by phase detection of the earth scan waveform with respect to a fixed reference. The reference is initailly set to coincide with the vehicle control axis. Therefore, whenever the vehicle attitude changes with respect to the local vertical, an error signal is generated by the sensor. This error signal is then directly proportional to the vehicle attitude error. By utilizing two of these sensors, one aligned to the roll axis, and one aligned to the pitch axis as shown below in figure 11, the horizontal reference plane of the vehicle with respect to the earth is established.

The IR sensor will yield a position error signal whenever the vehicle is misaligned with respect to the earth. However, certain natural phenomena will also affect the signal output. Clouds circling the earth are at a lower temperature than the earth itself. This results in a nonuniform earth pulse when the earth is scanned by the sensor. As a result, an erroruneous position signal will be obtained. Another source of error is the grazing sun. This occurs whenever the sun rises or sets over the horizon, and is in the scanner field of view. This condition does not produce a sufficient temperature to activate the sun gate, and appears as part of the

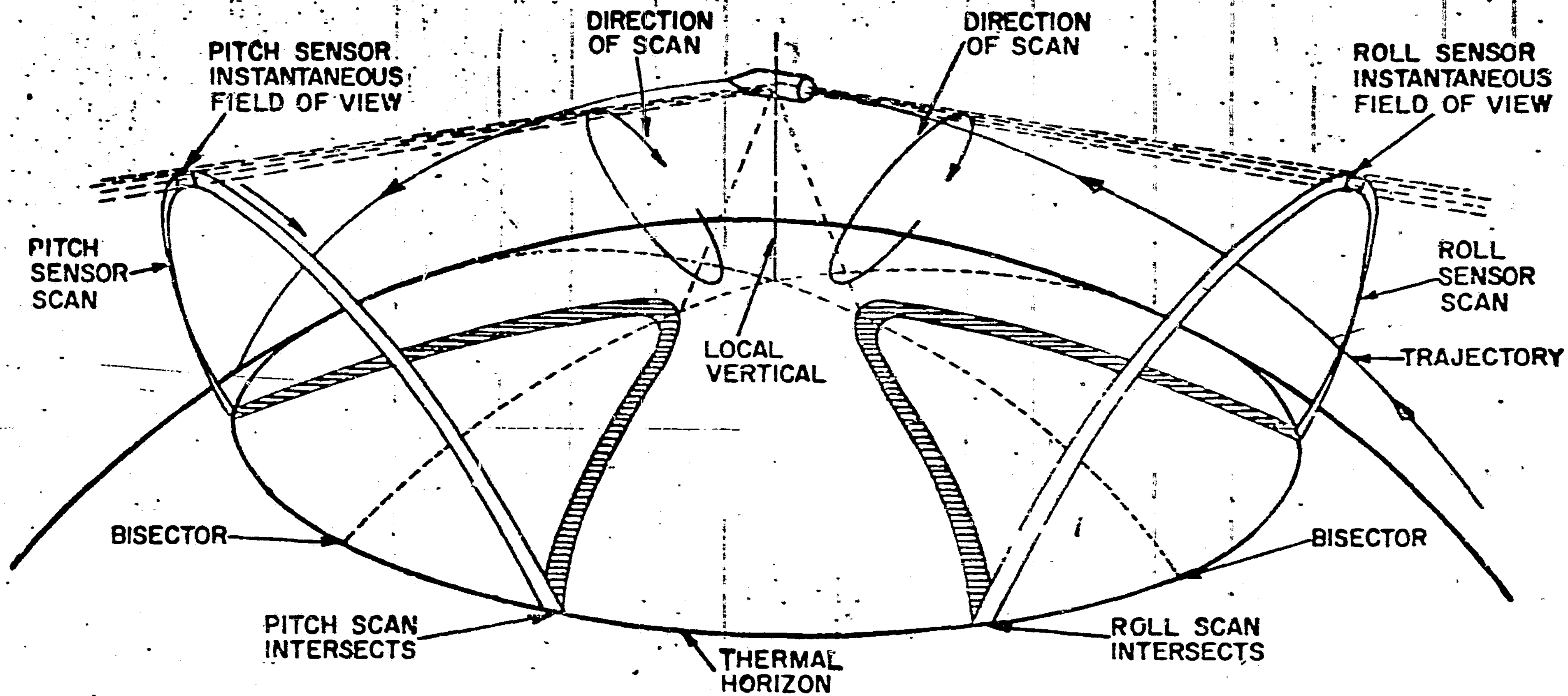
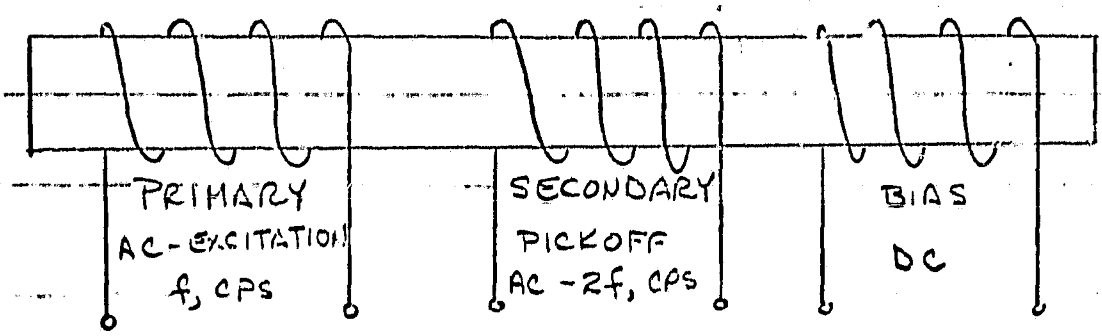



Figure 11 . INSTALLATION AND SCAN PATTERN, INFRARED HORIZON SENSOR

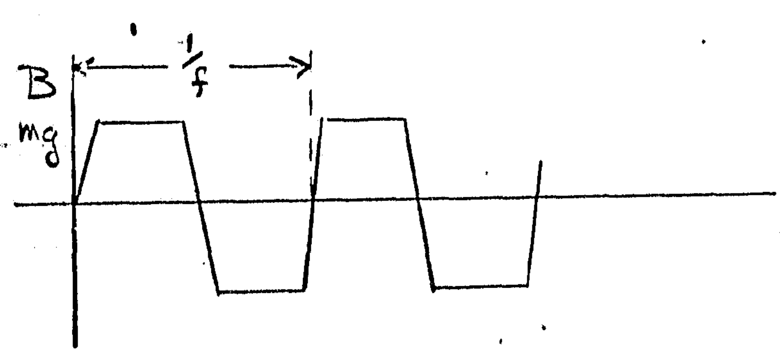
earth, again tiving a bad position signal.

The probability of these conditions appearing over the deorbit zone is small, and prior knowledge of the spacecraft performance will alert the ground controller to their presence.

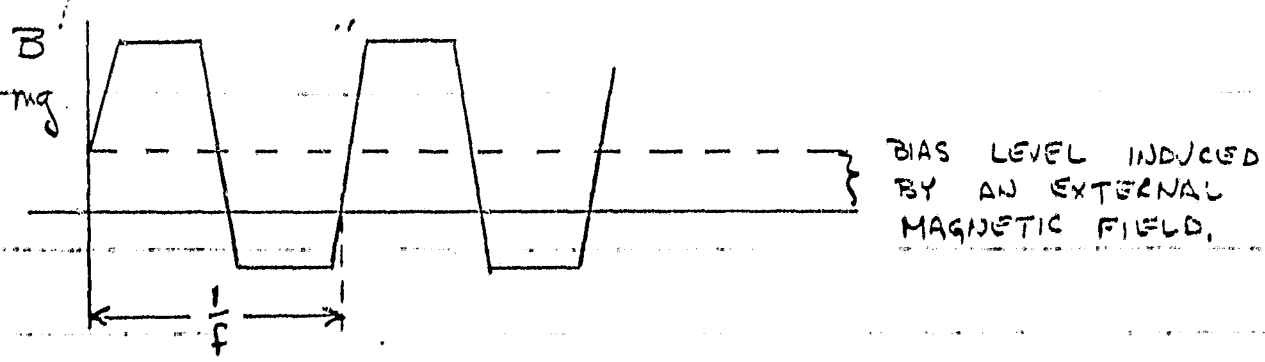
Magnetometer - The magnetometer is an electromagnetic device utilized to control the yaw axis of the spacecraft. It consists of 3 windings around a magnetic core, a primary winding, a secondary winding, and the bias winding. The primary winding is excited with an AC signal of sufficient amplitude to saturate the magnetic material. Because of the magnetic saturation, the




flux which is induced in the core will appear as a square wave of frequency, f.




However, a square wave with the zero crossing midway between the maximum and minimum amplitude as shown above has no even harmonic components. When a magnetic field, such as that of the earth's horizontal component, is directed longitudionally down the length of the magnetic core, ~~this~~ which has the effect of biasing the square wave excitation signal. As the square wave is biased so that the minimum (or maximum) amplitude tends toward zero, the even harmonics appear, such that a square wave as shown below has even and odd harmonics.




Therefore, the amplitude of the even harmonics is proportional to the bias field or the magnetic field. Thus a measurement of the magnetic field may be obtained by measuring the relative amplitude of the second harmonic. This is accomplished by tuning the secondary winding to the second harmonic. The signal from the secondary winding or pickoff may be amplified and rectified to yield a DC signal proportional to the magnetic field.

The bias winding maintains a constant magnetic field in the core. This is accomplished by applying a fixed DC current to obtain a known constant field. The bias field is useful for the cancellation of known stray disturbance fields and portions of the earth's magnetic field as on Biosatellite.

When controlling the yaw axis of the spacecraft, the vehicle is rotated by the propulsion system so as to align the magnetometer to give a zero output.

As shown in figure 12 , The magnetic error signal is null when the magnetometer is perpendicular to the resultant of the bias and horizontal component of earth's field. With proper bias applied to the magnetometer, the vehicle may be aligned in any attitude with respect to a known magnetic field, which will null out this field.

The scale factor of the magnetometer (volts/degree) is dependent upon the field being sensed. Therefore, the gain of the control loop will change accordingly, and the typical yaw switching lines shown in figure 13 are valid for a 400 milligauss field only. As the field decreases the slope of the switching lines will increase and the saturation level will therefore increase. As the slope approaches zero, the saturation level becomes increase-ingly important. However, for the range of the earth's field which is sensed by the Bias magnetometer (250-400 mg), the decrease of the saturation level will neither affect the acquisition or the limit cycle of the vehicle.

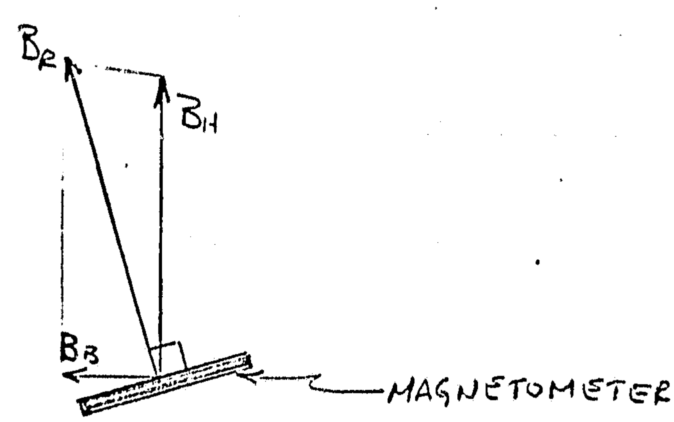



FIGURE 12 - MAGNETIC FIELD VECTORS

YAW SWITCHING LINES (400 MG FIELD)

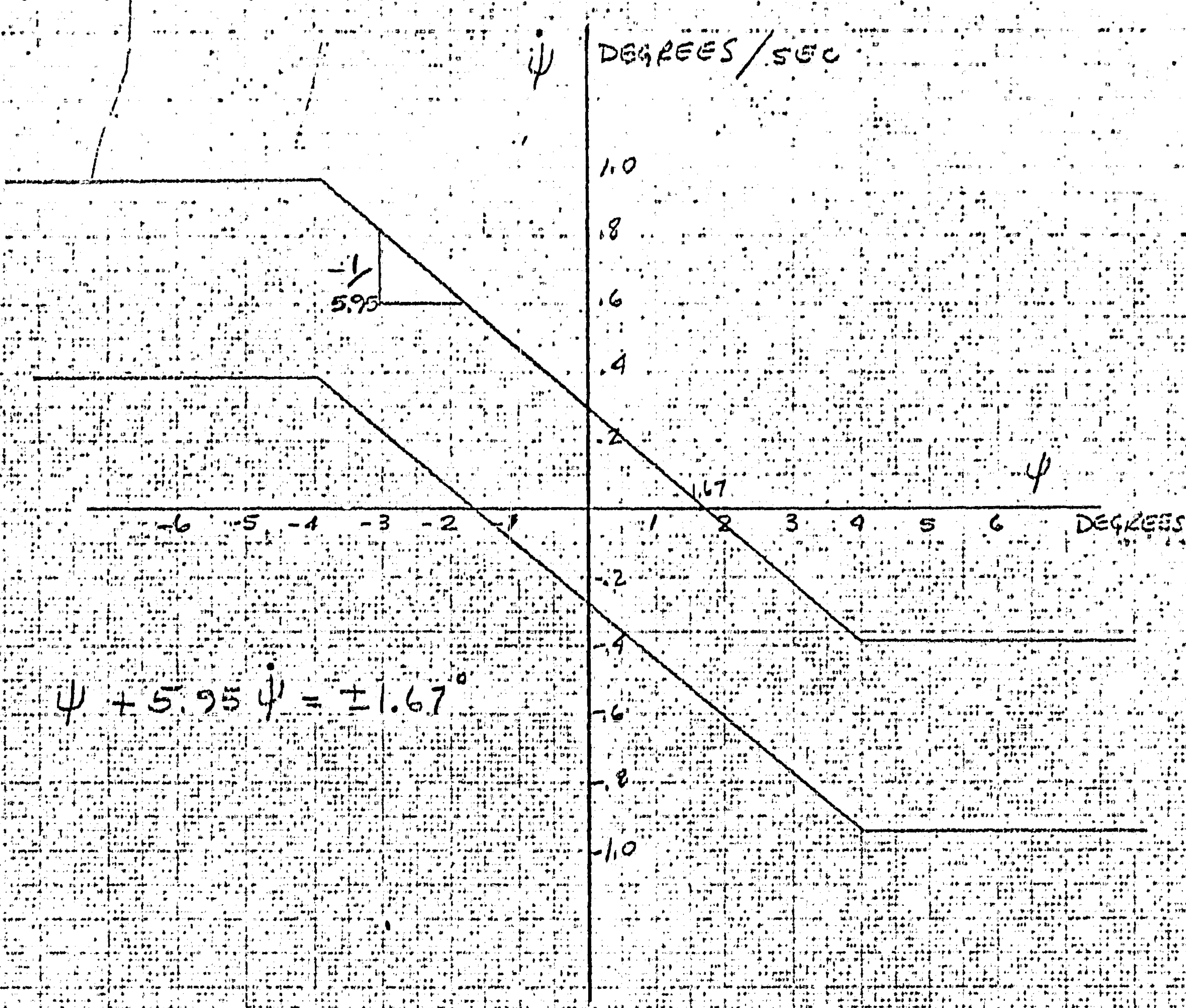



FIGURE 13

APPENDIX A

## Orbital Disturbance Forces

The rate control mode maintains the vehicle rates about each of the axes such that the forces on the spacecraft do not exceed $10^{-5}$ G. This rate limit has been previously calculated to be 0.286 degrees/sec. In space, with an absence of AN ATMOSPHERE AND a gravitational field, the disturbance forces (SUCH AS SOLAR PRESSURE) are indeed low. However, both translational and rotational forces do exist, and if allowed to react upon the vehicle unchecked, they would quickly induce an artificial gravitational field upon the vehicle, and invalidate the experiment. The forces which are known to exist are shown below:

Drag- 201 NM altitude (translational) -------- $0.11 \times 10^{-5}$ G
Exhausting Fluids (rotational)---------------$0.406 \times 10^{-5}$ G
Specimen (rotational,oscillatory)------------$0.220 \times 10^{-5}$ G

Except for the translational drag forces, all rotational forces may be contained with the attitude control system. Figure A1 shows how rapidly forces may build up, and after 10 days in orbit the exhausting fluids would build up such that the forces will exceed $10^{-3}$ G, much beyond the tolerable level. However, when rate control was utilized, the phase plane of figure A2 indicates how these disturbances may be nulled. A boiler exhausting fluids for 17 minutes will induce 3 axis rates on the vehicle, and in the pitch axis,

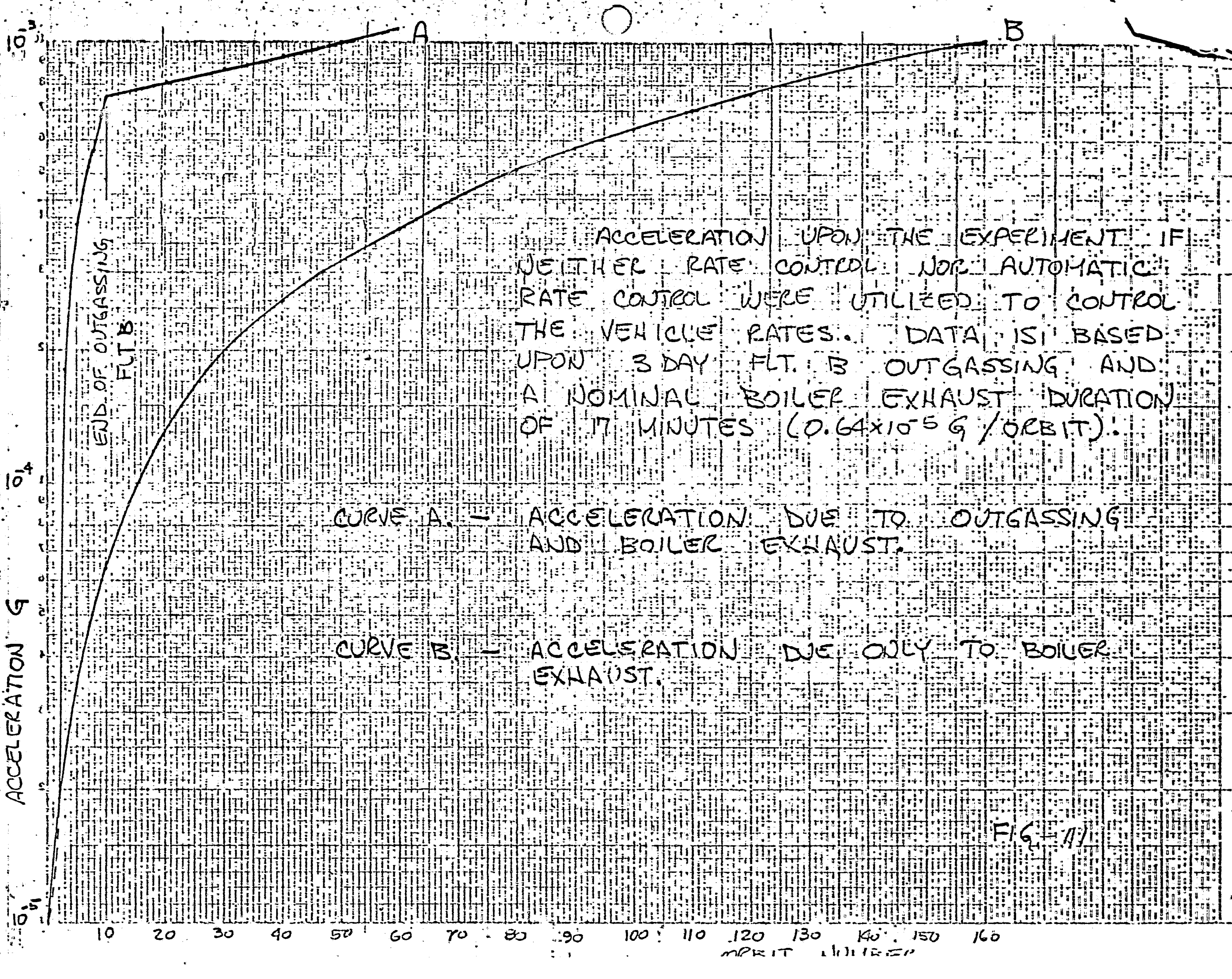

ACCELERATION UPON THE EXPERIMENT IF NEITHER RATE CONTROL NOR AUTOMATIC RATE CONTROL WERE UTILIZED TO CONTROL THE VEHICLE RATES. DATA IS BASED UPON 3 DAY FLT. B OUTGASSING AND A NOMINAL BOILER EXHAUST DURATION OF 17 MINUTES (0.64x10-5 G / ORBIT).
CURVE A. — ACCELERATION DUE TO OUTGASSING AND BOILER EXHAUST.
CURVE B. — ACCELERATION DUE ONLY TO BOILER EXHAUST.
END OF OUTGASSING FLT B
A
B
ACCELERATION G
10-3
10-4
10-5
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150
160

FIG-11

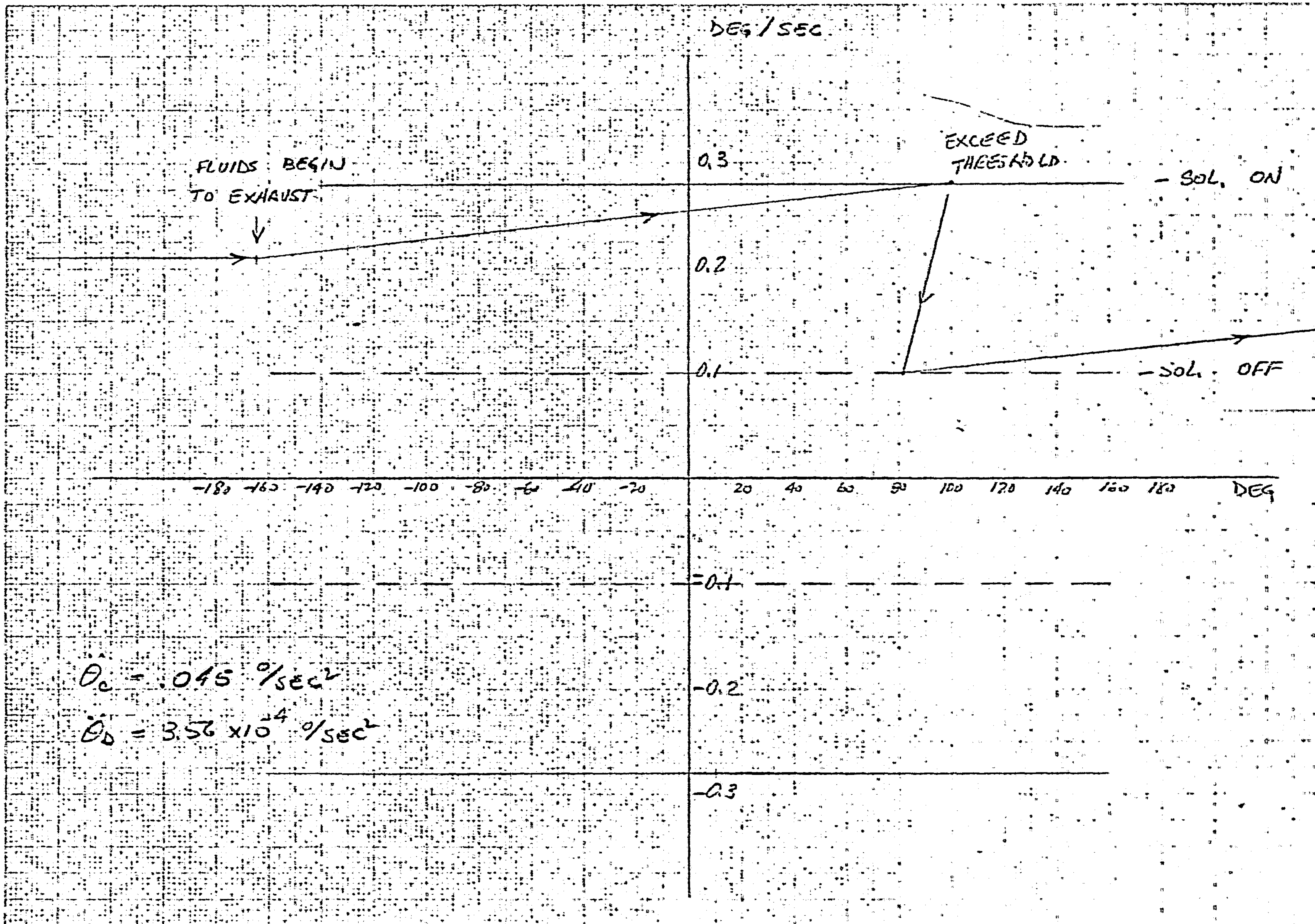



FIGURE A2 - RATE CONTROL PHASE PLANE

these rates will reach a manimum of 0.36 degrees/sec/orbit, if unchecked. To conserve fuel, hysteresis is built into the control. The control system hysteresis drives the rates far below the threshold, and since the disturbance is still present they immediately begin to build up again. Without hysteresis this would cause chatter in the system, and the fuel consumption during this mode of operation would increase, substantially.

The gas consumption of the system while operating in the rate mode was calculated to be approximately 4.8 lb. This considered THE worst case boiler disturbance torques (AS DETERMINED BY AN ANALYSIS OF THE BOILER SYSTEM) of 17 minutes duration per orbit, and hysteresis as shown. To obtain more accurate data based upon mission contingencies, a 3 axis hybrid simulation was generated which considered altitude variations, atmospheric density, guro performance as well as the boiler torques. The results are shown in table AI. The hand calcuations compared favorably with the computer simulation, since the computer predicted a manimum of 5.67 lb at 200 NM altitude.

Gas Consumption Calculation -

$\Delta\dot{\theta}$ = 0.186 deg/sec -between switching line "on" and hysteresis "off"

$\ddot{\theta}_D$= 3.56 x $10^{-4}$ deg/sec$^2$-disturbance acceleration from boiler

$\ddot{\theta}_C$= 0.045 deg/sec$^2$ -control acceleration

Duration of disturbance -17 min/orbit

time required to build up excessive rates $=\Delta\dot{\theta}/\ddot{\theta}_D$ = .186/3.56 x $10^{-4}$= 522 sec

number of control = 17min x 60sec/min / 522 sec = 1.96 = 2
periods per orbit

control thrust duration = $\Delta\dot{\theta}/\ddot{\theta}_C$ = .186/.045 = 4.13 sec

Pneumatic flow rate = $20.2 \times 10^{-4}$ lb/sec

Mission flow rate = $20.2 \times 10^{-4} \times 4.13 \times 3 \times 1/1.5 \times 24 \times 30$ = 4.8 lb/mission

= lb/sec x sec/orbit x nozzles x orbit/hours x hours/day x days/mission = lb/mission

## TABLE AI-GAS CONSUMPTION 4

| Altitude | Air Density | GAs Consumption | Comments |
|---|---|---|---|
| 115 | High | 1.25 lb-m | Minimum Altitude |
| 220 | High<br>Low | 1.54 lb-m<br>1.27 lb-m | Maximum Altitude |
| 135 | High | 1.94 lb-m | Maximum-Nominal gyros |
| 180 | Low | 0.89 lb-m | Minimum-Nominal gyros |
| 130 | High | 1.71 lb-m | Maximum-Degraded Gyros |
| 160 | High | 0.51 lb-m | Minimum-Degraded Gyros |
| 115 | High | 5.20 lb-m | Worst Case Boiler Torques |
| 220 | Low | 5.67 lb-m | Worst Case Boiler Torques |
| 115 | High | 1.87 lb-m | |

Deorbit Mode Phase Plane

The deorbit mode phase plane of figure A3 shows the typical performance of an uncrosscoupled single axis of the attitude control subsystem. Since the rates may never exceed 0.286 deg/sec, the acquisition time is dependent upon the magnitude of the position error. It may be calculated from equation A-1 below.

eq A-1 $$T_{acq} = 11.9 + \theta/.286 \quad \text{sec}$$

The acquisition time for the phase plane of figure A3 is 47 sec. It should be noted that the trajectory "walks up" the switching line (RETICENCE) prior to obtaining a stable limit cycle. This normally IS an inefficient means of acquiring the deadband. Since the fuel consumption and acquisition time of these systems are usually excessive. However, for this particular design, this is a secondary consideration. The torque/inertia ratio of 0.045 deg/sec$^2$ and the 0.286 deg/sec rate threshold must be maintained to meet the system requirements. To eliminate RETICENCE* ~~"walking"~~ at the sacrifice of either of these two requirements is unjustifiable, particularly since the trajectory is within the deadband limits before the RETICENCE ~~"walking"~~ begins, AND THE ATTITUDE ERROR SIGNAL IS WITHIN THE SUBSYSTEM REQUIREMENTS

* FURTHER DETAILS CONCERNING THE PERFORMANCE OF BANG BANG CONTROL SYSTEMS MAY BE OBTAINED FROM ASTIA DOCUMENT # 271221 — " SINGLE-AXIS REGULATION OF EXTRA ATMOSPHERIC VEHICLES."

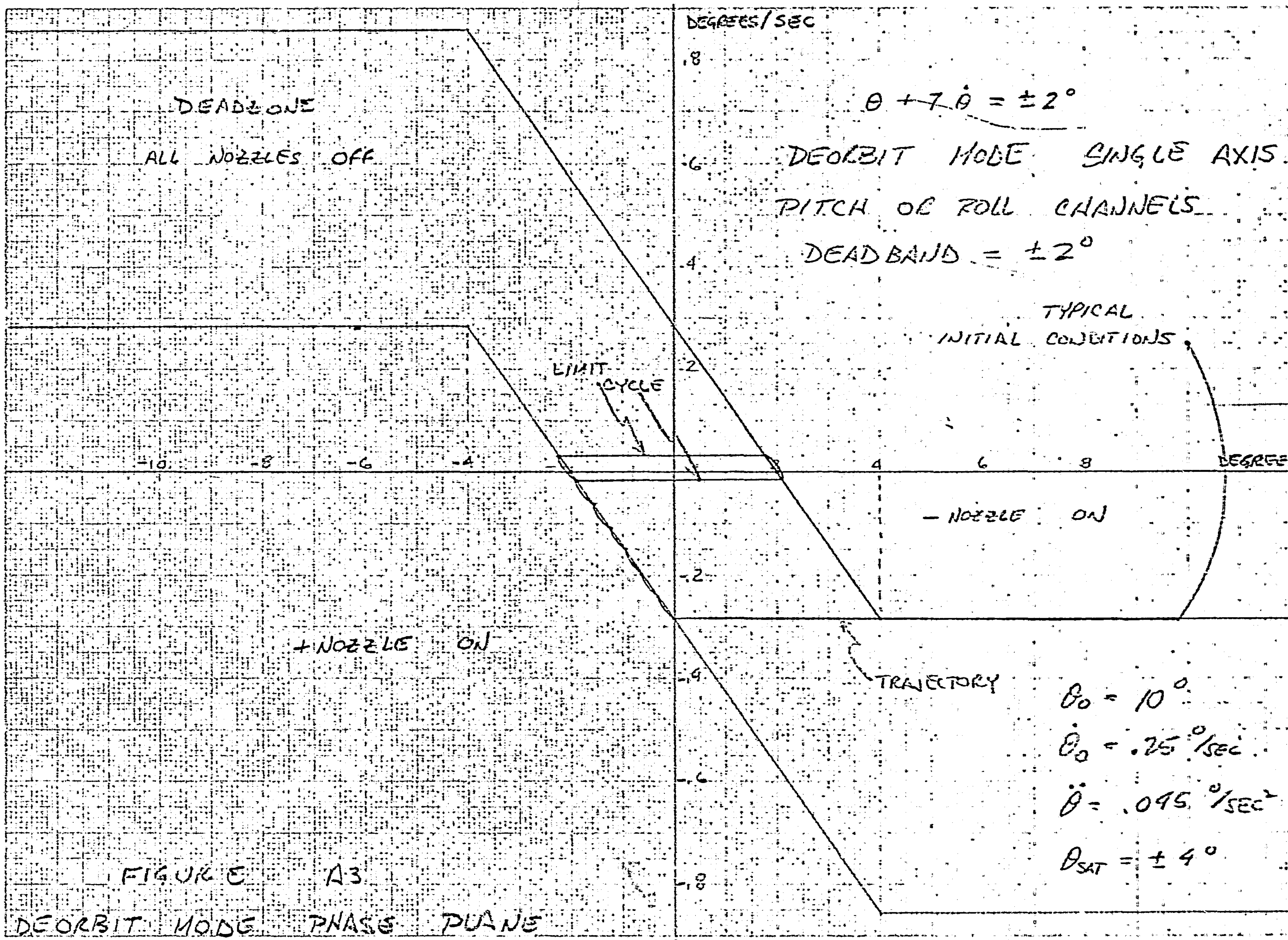



FIGURE A3
DEORBIT MODE PHASE PLANE

# Bibliography

1. E. T. Thomas - Rate and Acceleration Limits for New Biosatellite Configuration General Electric PIR # 8193-Y2-88

2. Carl Machover - Basics of Gyroscopes - John F. Rider Publisher, New York, 1960

3. Barnes Engineering Company - Operational Handbook for Model 13-133A Infrared Horizon Sensor, Barnes Engineering Co, 30 Commerce Road, Stamford, Connecticut

4. T. J. Smith - Biosatellite 30 Day ACS Design Review A/C Simulation - General Electric PIR # 8244-BIOS-882